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MEASUREMENT OF AERODYNAMIC HEAT TRANSFER TO A DEFLECTED TRAILING-EDGE FLAP ON A DELTA FIN IN FREE FLIGHT AT MACH NUMBERS FROM 1.5 TO 2.6

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MEASUREMENT OF AERODYNAMIC HEAT TRANSFER TO A DEFLECTED TRAILING-EDGE FLAP ON A DELTA FIN IN FREE FLIGHT AT MACH NUMBERS FROM 1.5 TO 2.6[1]

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SUMMARY

Aerodynamic-heating data were obtained at Mach numbers from 1.5 to 2.6 from a free-flight test on a sealed trailing-edge flap on a clipped 60° delta fin. Measurements were made on both sides of the control for deflections of 10° and 20° and for a point on the upper and lower surfaces of the fin just ahead of the hinge line. The control had a blunt trailing edge and a chord equal to 14.4 percent of the mean aerodynamic chord. The Reynolds number of the test varied from 11×10^{6} to 18.1×10^{6} based on the wing mean aerodynamic chord of 1.48 feet.

The heat-transfer coefficients expressed in dimensionless form as Stanton numbers are presented as a function of free-stream Mach number and indicate that, in general, the heat-transfer coefficients on the windward side for a flap deflection of 20° were about 2.5 times those of the measurements made on the fin while those on the leeward side were one-third those of the fin. The heat-transfer coefficients for a flap deflection of 10° were about 1.75 and 0.6 times the fin coefficients for the windward and leeward sides, respectively. Theory for turbulent flow on a flat plate was in good agreement with the coefficients when the theory was based on the estimated local flow conditions and length from the fin leading edge to the measurement station.

INTRODUCTION

The study of aerodynamic heating at supersonic speeds has, in the past, been investigated for very simple shapes for which heat-transfer theory was applicable. However, because of the progressively increasing Mach number ranges of research airplanes and advanced military aircraft, the need for information on aerodynamic heating of various components has

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[1]Supersedes recently declassified NACA Research Memorandum L57B20 by Leo T. Chauvin and James J. Buglia, 1957.

become urgent. Consequently, tests have been conducted by the Langley Pilotless Aircraft Research Division to measure the aerodynamic heat transfer to an aircraft canopy, a delta wing at zero angle of attack, a delta wing at angle of attack, and a deflected control surface. The highlights of this program were presented in reference 1, and detailed information on several phases of the program has since been published in references 2 to 4. The present investigation gives in greater detail the information obtained from the study of deflected control surfaces.

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Severe aerodynamic heating as compared with that encountered by the wing may be experienced when a control is deflected. With the assumption that the flow does not separate, the heat transfer on the windward side would be higher because of increased pressure and possibly because of thinning of the boundary layer on the control surface; on the other hand, if separation does occur and the boundary layer is turbulent, the possibility of still higher heat-transfer rates is suggested by theory in reference 5 and by experiment in reference 6. The resulting temperature difference across the control could be considerable; the heating rate between the windward and leeward sides of the control could cause severe thermal stresses in the structure of the control.

These considerations necessitated the determination of the heating rates and also the comparison with available theories to show if these heating rates could be predicted. A two-stage rocket-propelled research vehicle was flight tested to determine the heat transfer to a control surface deflected 10° and 20°. The control surface was a sealed trailing-edge flap extending the full span on a 60° clipped delta wing and had a chord equal to 14.4 percent of the mean aerodynamic chord and a blunt trailing edge. Heat-transfer measurements were made on both sides of the control and for a point on the upper and lower surfaces of the fin ahead of the hinge line. The Mach number range of the test was 1.5 to 2.6 with corresponding free-stream Reynolds number of 11.0×10^{6} to 18.1×10^{6} based on wing mean aerodynamic chord of 1.48 feet. The flight test was conducted at the Langley Pilotless Aircraft Research Station at Wallops Island, Va.

SYMBOLS

M	Mach number
R	Reynolds number
P_{∞}	pressure, lb/sq in.
ρ	density, slugs/cu ft

T	temperature, $^{\circ}R$
N_{St}	Stanton number, $h/c_p \rho V g$
N_{Pr}	Prandtl number
X	station of fin section, percent chord
Y	ordinate of fin section, percent chord
C_f	skin-friction coefficient
h	aerodynamic heat-transfer coefficient, Btu/sec-sq ft-$^{\circ}F$
c_p	specific heat of air, Btu/slug/$^{\circ}F$
V	velocity, ft/sec
g	gravitational acceleration, ft/sec^2
τ	skin thickness, ft
C_w	specific heat of wall, Btu/lb/$^{\circ}F$
ρ_w	specific weight of wall, lb/cu ft
RF	temperature recovery factor
t	time, sec
δ	control deflection, deg
f	arbitrary function

Subscripts:

∞	free-stream condition
w	pertaining to wall
aw	adiabatic wall
v	local conditions outside the boundary layer
so	isentropic stagnation

MODEL, INSTRUMENTATION, AND TEST

Model

A sketch of the model showing the arrangement of the fins is given in figure 1. The four stabilizing fins with controls at fixed deflections were mounted to the cylindrical section of the fuselage at the rear of the body. Two fins in one plane had their controls deflected $\pm 10^{\circ}$. The two fins in the opposite plane had their controls fixed at $\pm 20^{\circ}$. The controls were deflected in such a manner that the rolling moment created by the control deflected $\pm 10^{\circ}$ opposed the rolling moment created by the control deflected $\pm 20^{\circ}$. The clipped delta fin was swept back 60° and had an aspect ratio of 0.68, based on the exposed area of one fin. Details of the fin are presented in figure 2. The ordinates of the fin airfoil section are given in the figure for zero control deflection. The full-span sealed control had a constant chord, 14.4 percent of the mean aerodynamic chord of the fin. The control ordinates are those of the fin section bent to the required angle and faired smooth to the wing at the hinge line.

The construction of the fin consisted of an aluminum-alloy plate, bent to the required control deflection, with mahogany fill bonded to it and an overlay of Inconel sheet 1/32 inch thick bonded to the wood. In order to reduce heat-conduction effects, the wood near the thermocouples was removed as shown in figure 2 and a nylon strip forming the trailing edge was used to insulate the aluminum core and the two surfaces of the outer skin of the control from each other.

The model was propelled by a two-stage propulsion system, the first stage being two 6.25-inch ABL Deacon rocket motors firing simultaneously, and the second stage consisting of another Deacon rocket motor carried within the cylindrical shell of the model. A photograph of the model showing the booster arrangement is shown in figure 3.

Instrumentation

The NACA telemetering system which was carried in the nose of the fuselage transmitted measurements of temperatures and of longitudinal acceleration. Fin and control temperatures were commutated at the rate of one reading every 0.2 second. The temperatures were measured at 12 points on each of the two fins, one of which had its control deflected 10° and the other 20°. Thermocouples were located on both sides of the control surface and on the wing directly ahead of the hinge line as shown in figure 2. The thermocouples were made of no. 30 iron-constantan wire fused to the inside of the Inconel skin. Prior to the test, thermocouple no. 7 failed. The accuracy of the measured temperatures is believed to be within $\pm 10^{\circ}$ F; however, a more complete discussion of the method of temperature-telemetering technique employed by the Langley Pilotless Aircraft Research Division may be found in reference 7.

The rolling velocity of the model was measured during the flight by means of a polarized telemeter antenna signal.

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In addition to the velocity obtained from the integration of the longitudinal acceleration, velocity data were obtained by means of CW Doppler radar set, and altitude and flight-path data were measured by NACA modified SCR-584 tracking radar. Atmospheric and wind conditions were measured by means of radiosondes launched near the time of flight and tracked with Rawin set AN/GMD-1A.

Test

The model was launched at an elevation angle of 70°. The booster accelerated the model to a Mach number of 1.53, where it drag-separated at burnout. The model then coasted upwards for a predetermined time at which the rocket motor within the model ignited and accelerated the model to a Mach number of 2.6 at 19.8 seconds. As a result of the control settings, the model rolled, and from the measurements of rolling velocity it was estimated that the fin angle of attack at the midspan station was approximately 1°. Shortly after 20 seconds structural failure of the fins was observed.

Atmospheric conditions for the test are given in figure 4 as a function of time of flight. These data were reduced from radiosonde observations made close to the time of the flight test which were then correlated with the actual flight by means of the position radar (SCR-584) which yielded altitude time history.

Mach number and Reynolds number per foot which were reduced from radiosonde observations and from velocity data from CW Doppler radar and integrated longitudinal accelerometer are presented in figure 5.

DATA REDUCTION

The free-stream Stanton number N_{St} can be determined from the following relation:

$$N_{St} = \frac{h}{\left(c_p \rho V g\right)_v} = \frac{1}{\left(c_p \rho V g\right)_v} \frac{\tau_w \rho_w C_w}{\left(T_{aw} - T_w\right)} \frac{dT_w}{dt}$$

The properties of the wall are known and the rate of change of the wall temperature is the slope of the measured time history of the skin temperature. The properties of the air are obtained from radiosonde observation, and velocity was measured by the Doppler radar or calculated from measurements of acceleration. In order to obtain the temperature difference $T_{aw} - T_w$, it is first necessary to know the adiabatic wall temperature T_{aw} which is obtained from the definition of recovery factor

$$RF = \frac{T_{aw} - T_v}{T_{so} - T_v}$$

A turbulent recovery factor $RF = N_{Pr}^{1/3}$ based on wall temperature was assumed. The assumption is considered reasonable from a consideration of the magnitude of the test Reynolds number and a tunnel test on a similar wing at approximately the same Reynolds number (ref. 4).

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The equation for Stanton number N_{St} is valid for this test because heat losses due to conduction and radiation were estimated and found to be negligible compared with the large heat flow into the wing.

The purpose of evaluating the Stanton number as a function of free-stream conditions is to make possible a direct comparison between the heat-transfer rates to the fin and to the deflected controls. However, in order to compare the experimental data with theory, the experimental values of Stanton number must be based on local conditions or else the theoretical values of N_{St} which will be a function of local conditions must be converted to free-stream conditions. The latter approach was taken. Either approach requires the estimation of the local conditions. Local conditions on the fin were evaluated by using linear theory to calculate the pressure coefficient on the fin and then using two-dimensional shock and expansion theory for the windward and leeward sides of the control, respectively. With the local conditions known, a theoretical Stanton number (ref. 8) based on free-stream conditions and modified according to reference 9 and distance from the leading edge to the measurement station was obtained in the following manner:

$$\left(N_{St}\right)_{v,theory} = 0.6C_f$$

where

$$C_f = f\left(M_v, R_v, \frac{T_w}{T_v}\right)$$

$$\left(N_{St}\right)_{\infty,theory} = \left(N_{St}\right)_{v,theory} \frac{\left(\rho c_p V\right)_{v,estimated}}{\left(\rho c_p V\right)_{\infty,measured}}$$

RESULTS AND DISCUSSION

From the measurements of skin-temperature time histories given in figure 6, Stanton numbers N_{St} were reduced as described in the section entitled "Data Reduction" for times between 18.25 seconds to 19.8 seconds corresponding to Mach numbers between 1.5 to 2.6. Heat-transfer data were not reduced prior to 18.25 seconds because of loss in accuracy for the low heating rates. Figures 7(a) and 7(b) show the values of N_{St} plotted as a function of free-stream Mach number for $\delta = 10^{\circ}$ and 20°, respectively. By presenting N_{St} based on free-stream condition, a direct comparison of the heat-transfer coefficient can be made for various stations on the wing.

L
8
2
4

The accuracy of the heat-transfer measurements was estimated according to the method presented in the appendix of reference 10 and is discussed in this section to provide a basis for the analysis of the data. The estimates of the accuracy showed that at $M = 1.5$ the expected accuracy was of the order of 14 percent, improving rapidly with an increase in Mach number to values of 8 and 3 percent at Mach numbers of 1.8 and 2.6, respectively, due mainly to the increase in $(T_{aw} - T_w)$. It is to be noted that these estimates do not consider any effects of bending of the wing on the heat-transfer measurements.

The close agreement between the N_{St} values for the fin ahead of the hinge line (station 1) in figures 7(a) and 7(b) is significant in that it shows that the angle of attack of the fin due to roll had little effect on the heat-transfer measurements. It was estimated that the rolling of the model would result in an angle of attack of less than 1° for this station.

For $\delta = 20^{\circ}$ (fig. 7(b)) the measured Stanton number on the windward side of the control for Mach numbers greater than 1.8 suggests a spanwise increase in heat transfer of approximately 15 percent from stations 2 to 3 which are located at 29.7 percent of the control chord, while for stations 4, 5, and 6, located at 76.7 percent chord, no spanwise effect is discernible. Chordwise variations of heat transfer at the 50-percent-span stations were about 10 percent (stations 2 and 5) while the tip stations (3 and 6) at 85-percent span showed no effect. This may result from the differences between local flow conditions at the tip and at the 50-percent-span stations. The leeward side of the control showed no spanwise effect on the heat transfer measured; however, a chordwise effect of less than 10 percent is indicated for the 50-percent- and 85-percent-span stations. Heat-transfer data for Mach numbers 1.5 to 1.8 for the windward side are presented even though they represent a condition where the shock is theoretically detached from the wedge and data were reduced by assuming the free-stream static temperature in computing the adiabatic wall temperature.

Data for the control deflected 10° are presented in figure 7(a). The chordwise and spanwise effects for this control are less severe than for $\delta = 20^{\circ}$, as the only indicated effect is a 10-percent spanwise variation in the heat-transfer coefficient for stations 2 and 3. Although some variations are shown for the other stations, they are of the order of the accuracy discussed previously.

The Stanton numbers for $\delta = 20^{\circ}$ and $\delta = 10^{\circ}$ indicate that on the average the heat transfer on the windward side of the 20° control is about 2.5 times that of the fin immediately ahead of the hinge line while on the leeward side the average was about one-third that of the fin. The data for $\delta = 10^{\circ}$ show that Stanton numbers for the windward side are about 1.75 times those of the fin while those for the leeward side are about 0.6 those of the fin.

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The two theory curves shown in figure 7 for stations 3 and 4 are for flat-plate turbulent heat transfer as obtained from reference 8, the theory in this case having been determined for the estimated local conditions and the length from the leading edge of the fin to the measurement station. The theory is representative of that for the other points on the control at that particular span as the change in length in the Reynolds number is small. The theory for station 1 is for the points on the wing ahead of the hinge line based on length from the leading edge to station 1. Good agreement is shown for the windward side for both control deflections. The spanwise effect predicted by the theory is of the order of that shown experimentally for stations 2 and 3. Theory for the leeward side for $\delta = 10^{\circ}$ is in good agreement with the data. However, for $\delta = 20^{\circ}$ a greater chordwise effect is shown than would be indicated by theory. The theory for stations in a chordwise direction on the control showed essentially no change. For this reason only the theory for the 22.3- and 85.6-percent-span stations is presented.

CONCLUSIONS

An experimental free-flight investigation to determine the heat-transfer coefficients for a full-span sealed trailing-edge control surface deflected 10° and 20° mounted on a 60° clipped delta fin and having a control chord 14.4 percent of the mean aerodynamic chord has been completed for a Mach number range of 1.5 to 2.6. From this test, the following conclusions can be made:

1. Heat-transfer coefficients on the windward side of the 20° deflected flap were on the average approximately 2.5 times the average of the measurements made on the fin ahead of the hinge line while the average for the leeward side was 0.3 that of the fin measurement. For the flap deflected 10° the average heat-transfer measurement was 1.7 times that of the fin for the windward side and 0.6 for the leeward side.

2. Flat-plate turbulent heat-transfer theory based on estimated local conditions and distance from the fin leading edge to the measurement station predicted the heat transfer with good accuracy.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., February 7, 1957.

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Ordinates of Nose

l	r
0	.192
.068	.208
.139	.240
.547	.336
1.40	.527
5.48	1.42
11.16	2.45
16.30	3.14
22.40	3.68
28.50	4.00

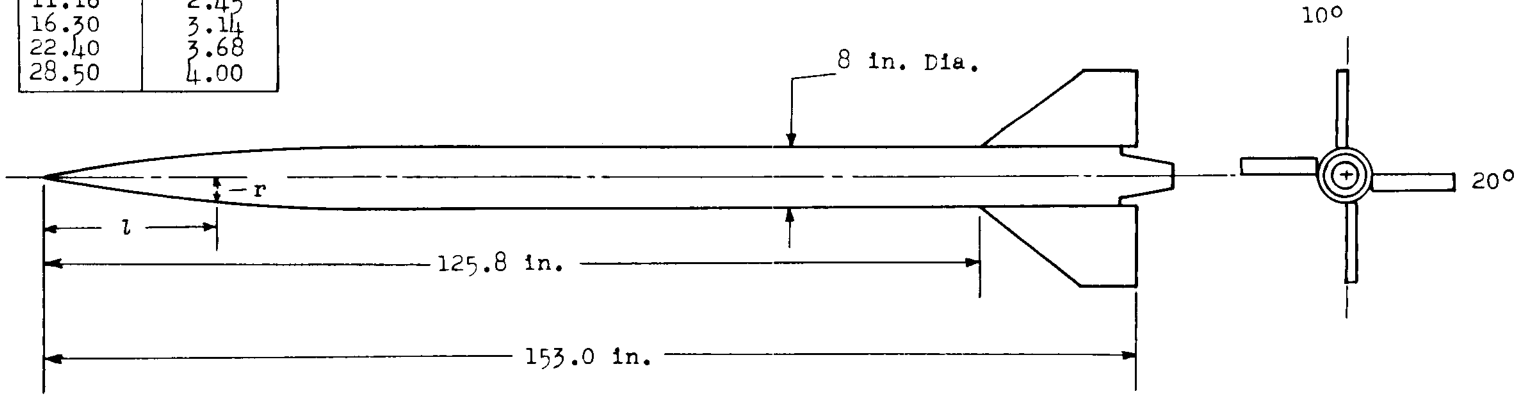

Figure 1.- General configuration of test vehicle. All dimensions are in inches unless otherwise noted.

Fin ordinates, Per cent chord

X	Y
0	0
.005	.0039
.008	.0047
.013	.0061
.025	.0086
.050	.0119
.075	.0139
.100	.0161
.150	.0196
.200	.0223
.250	.0245
.300	.0263
.350	.0276
.400	.0284
.450	.0289
.500	.0287
.550	.0281
.600	.0269
.650	.0254
.700	.0236
.750	.0214
.800	.0190
.850	.0165
.900	.0139
.950	.0115
1.000	.0088

Thermocouple Numbers

Sta.	$\delta = -20^\circ$	$\delta = -10^\circ$	$\delta = +10^\circ$	$\delta = +20^\circ$
1	17	9	3	23
2	16	11	4	22
3	14	7	2	20
4	18	12	6	24
5	15	10	5	21
6	13	8	1	19

Station Locations

Station	Distance from leading edge, in., $\delta = 10^\circ$ and $\delta = 20^\circ$
1	12.8
2	13.84
3	7.84
4	20.0
5	15.02
6	9.02

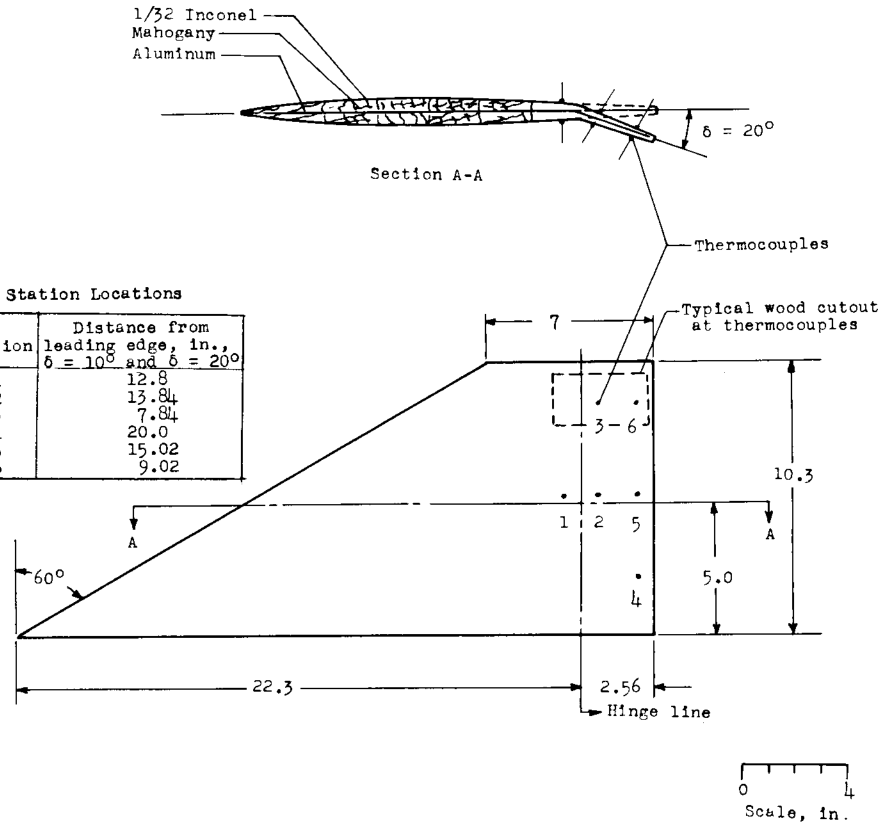

Figure 2.- Sketch of fin showing construction and location of thermocouples. All dimensions are in inches.

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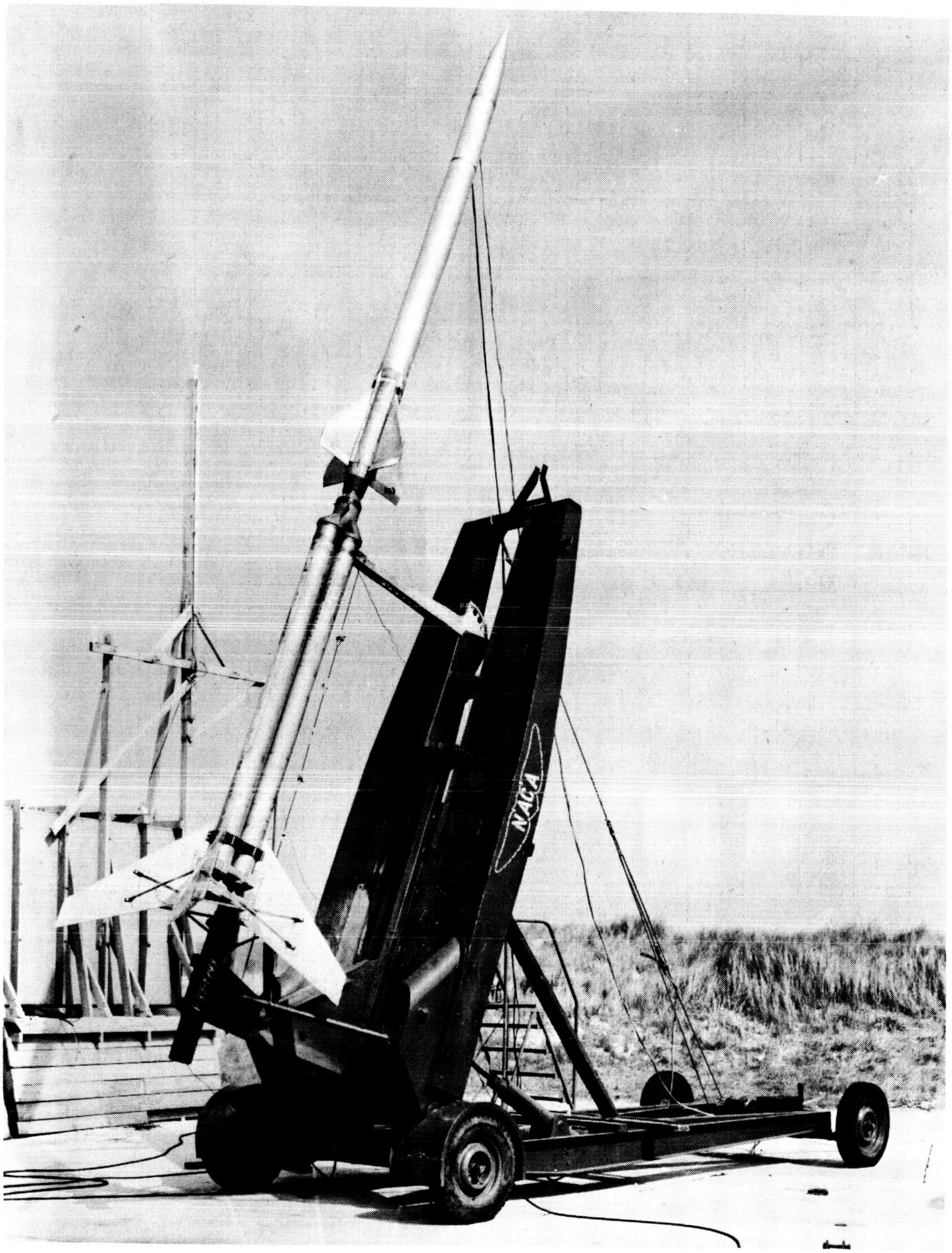


Figure 3.- Photograph of model in launching position. L-89538

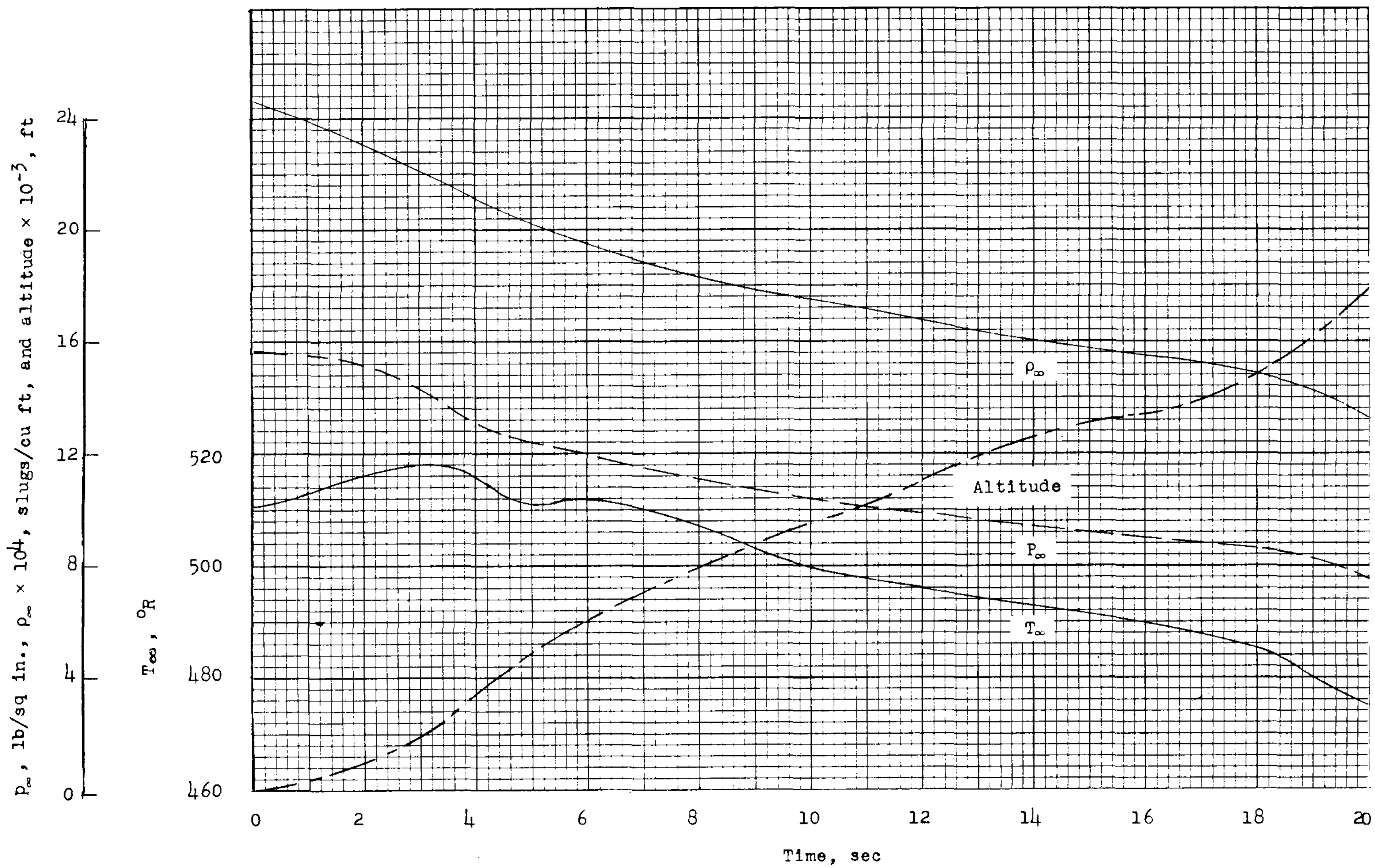


Figure 4.- Time histories of conditions related to model flight.

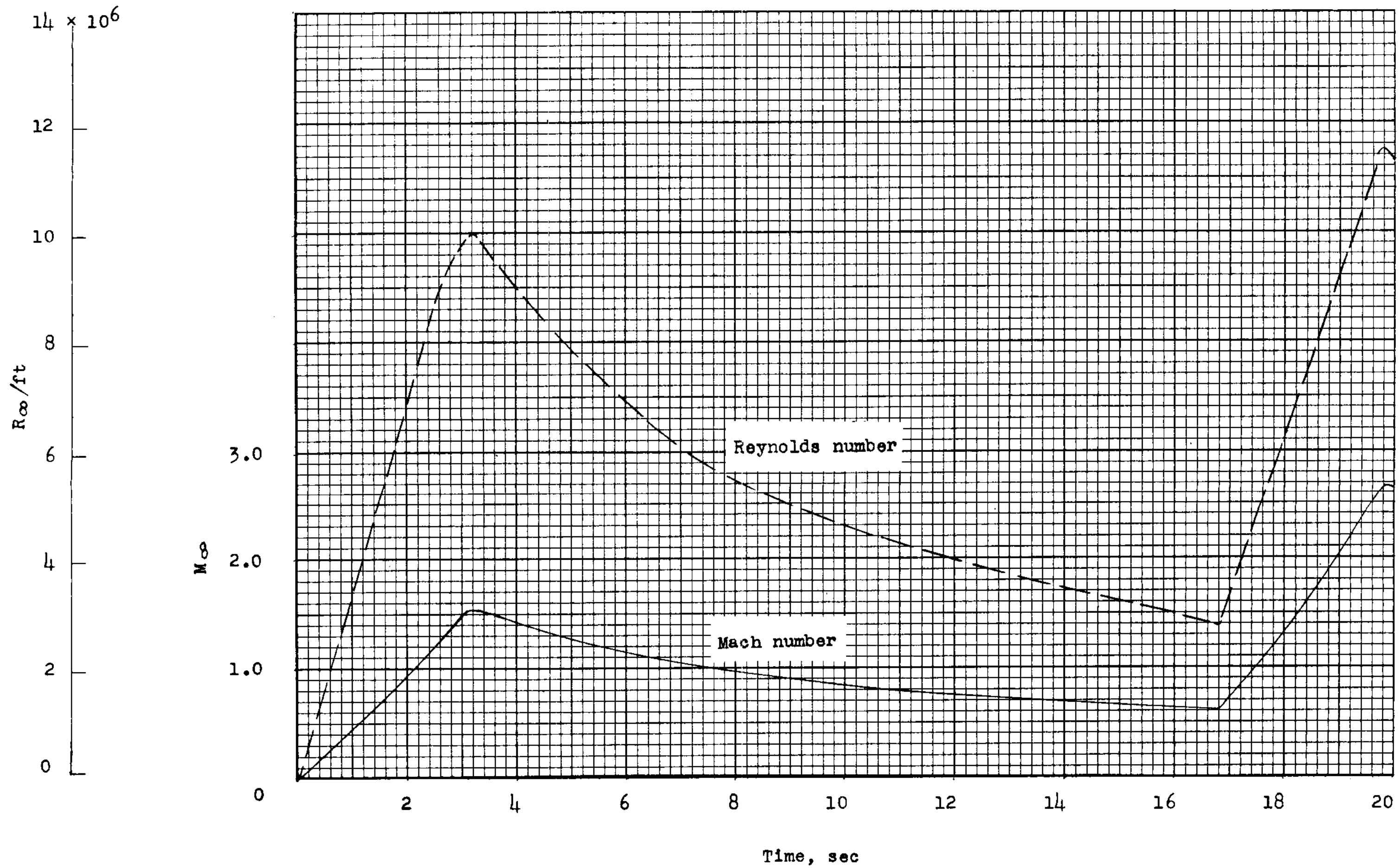
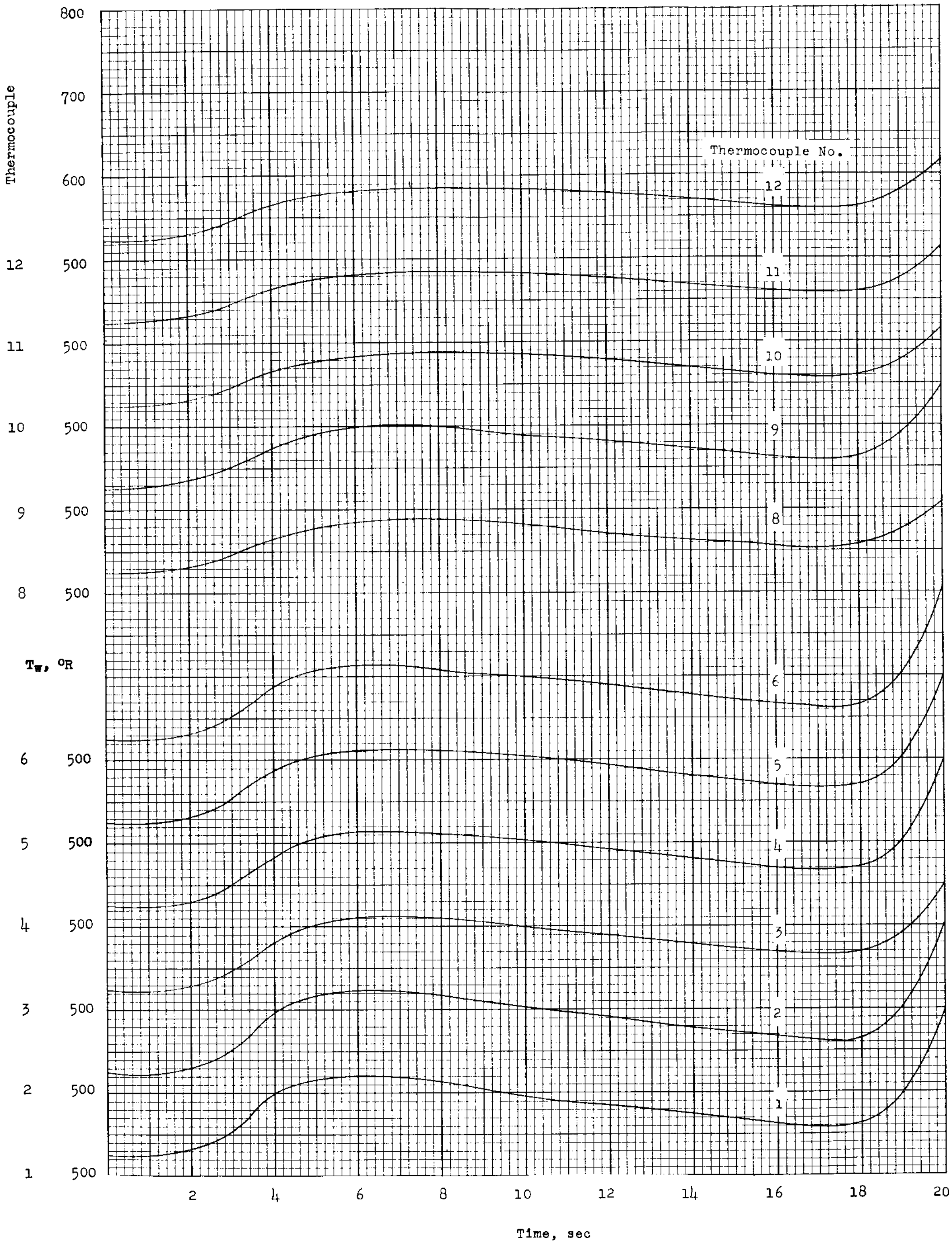


Figure 5.- Time histories of free-stream Mach number and Reynolds number per foot.



(a) $\delta = 10^{\circ}$.

Figure 6.- Time histories of measured temperatures.

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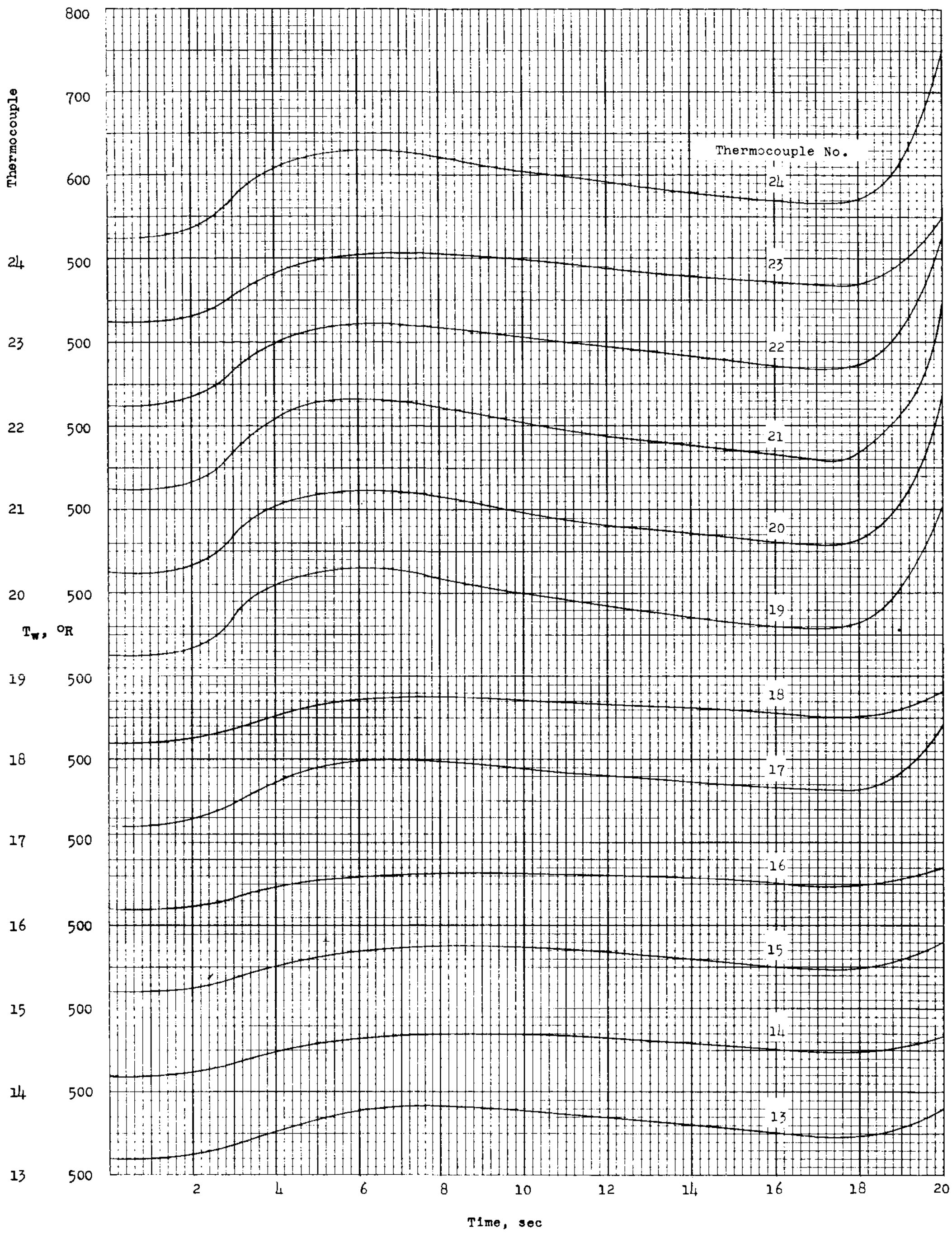

(b) $\delta = 20°$.

Figure 6.- Concluded.

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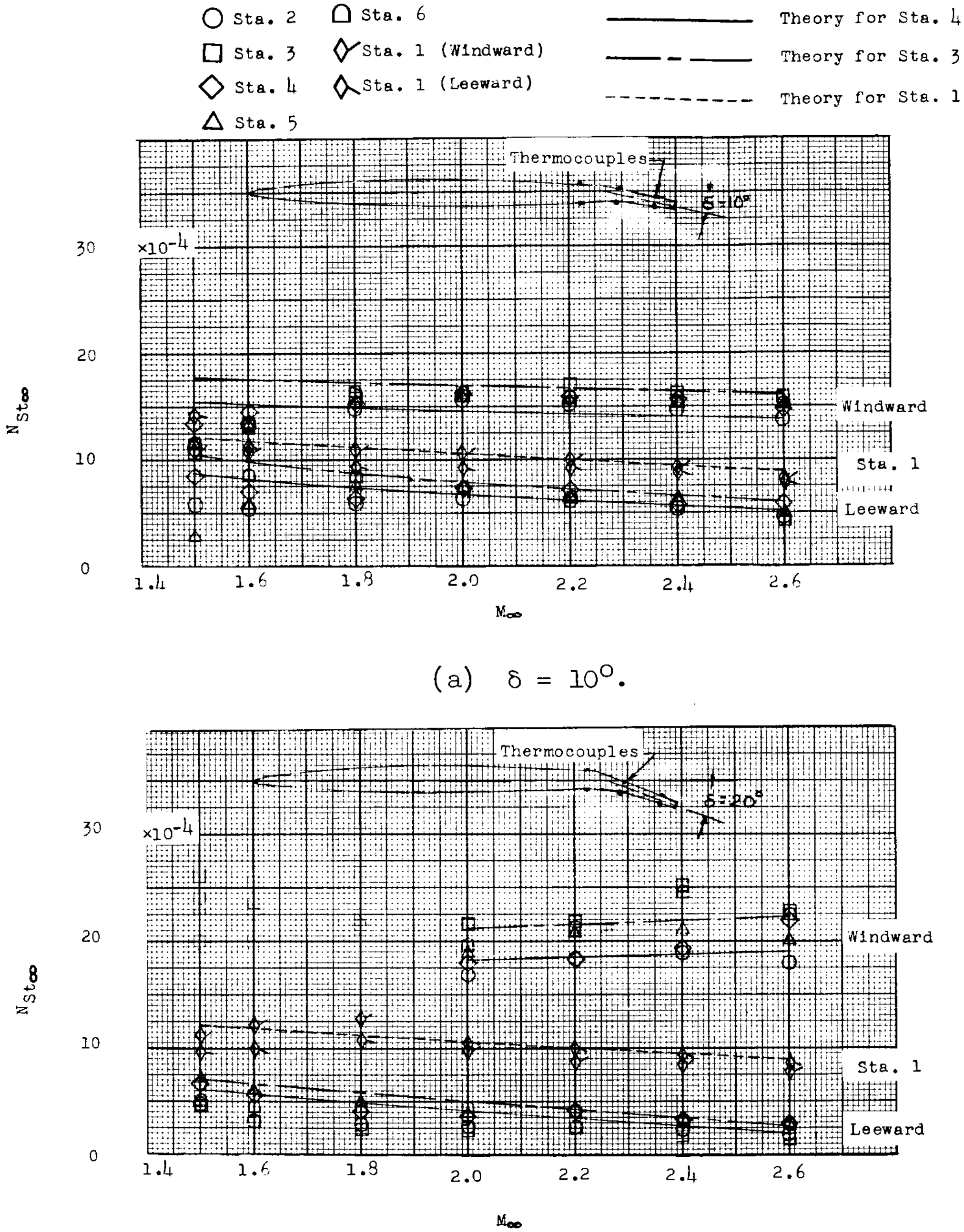

(a) $\delta = 10^{\circ}$.

(b) $\delta = 20^{\circ}$.

Figure 7.- Heat transfer for deflected control surfaces and fin.